4. The jet engine (fig. 1-2), although appearing so different from the piston engine-propeller combination, applies the same basic principles to effect propulsion. As shown in fig. 1-3, both propel their aircraft solely by thrusting a large weight of air backwards.

5. Although today jet propulsion is popularly linked with the gas turbine engine, there are other types of jet propelled engines, such as the ram jet, the pulse jet, the rocket, the turbo/ram jet, and the turbo-rocket.

**PRINCIPLES OF JET PROPULSION**

6. Jet propulsion is a practical application of Sir Isaac Newton's third law of motion which states that, 'for every force acting on a body there is an opposite and equal reaction.' For aircraft propulsion, the 'body' is atmospheric air that is caused to accelerate as it passes through the engine. The force required to give this acceleration has an equal effect in the opposite direction acting on the apparatus producing the acceleration. A jet engine produces thrust in a similar way to the engine/propeller combination. Both propel the aircraft by thrusting a large weight of air backwards (fig. 1-3), one in the form of a large air slipstream at comparatively low speed and the other in the form of a jet of gas at very high speed.

7. This same principle of reaction occurs in all forms of movement and has been usefully applied in many ways. The earliest known example of jet reaction is that of Hero's engine (fig. 1-4) produced as a toy in 120 B.C. This toy showed how the momentum of steam issuing from a number of jets could impart an equal and opposite reaction to the jets themselves, thus causing the engine to revolve.

8. The familiar whirling garden sprinkler (fig. 1-5) is a more practical example of this principle, for the mechanism rotates by virtue of the reaction to the water jets. The high pressure jets of modern firefighting equipment are an example of jet reaction; for often, due to the reaction of the water jet, the hose cannot be held or controlled by one fireman. Perhaps the simplest illustration of this principle is afforded by the carnival balloon which, when the air or gas is released, rushes rapidly away in the direction opposite to the jet.

9. Jet reaction is definitely an internal phenomenon and does not, as is frequently assumed, result from the pressure of the jet on the atmosphere. In fact, the jet propulsion engine, whether rocket, athodyd, or turbo-jet, is a piece of apparatus designed to accelerate a stream of air or gas and to expel it at high velocity. There are, of course, a number of ways of

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Fig. 1-2 A Whittle-type turbo-jet engine.

Fig. 1-3 Propeller and jet propulsion.
the former is preferred, since by lowering the jet velocity relative to the atmosphere a higher propulsive efficiency is obtained.

METHODS OF JET PROPULSION

10. The types of jet engine, whether ram jet, pulse jet, rocket, gas turbine, turbo/ram jet or turbo-rocket, differ only in the way in which the 'thrust provider', or engine, supplies and converts the energy into power for flight.

11. The ram jet engine (fig. 1-6) is an athodyd, or aero-thermodynamic-duct to give it its full name. It has no major rotating parts and consists of a duct with a divergent entry and a convergent or convergent-divergent exit. When forward motion is imparted to it from an external source, air is forced into the air intake where it loses velocity or kinetic energy and increases its pressure energy as it passes through the diverging duct. The total energy is then increased by the combustion of fuel, and the expanding gases accelerate to atmosphere through the outlet duct. A ram jet is often the power plant for missiles and target vehicles, but is unsuitable as an aircraft power plant because it requires forward motion imparting to it before any thrust is produced.

12. The pulse jet engine (fig. 1-7) uses the principle of intermittent combustion and unlike the ram jet it can be run at a static condition. The engine is formed by an aerodynamic duct similar to the ram jet but, due to the higher pressures involved, it is of more robust construction. The duct inlet has a series of inlet 'valves' that are spring-loaded into the open position. Air drawn through the open valves passes into the combustion chamber and is heated by the burning of fuel injected into the chamber. The resulting expansion causes a rise in pressure, forcing the valves to close, and the expanding gases are
Fig. 1-9-1 Mechanical arrangement of gas turbine engines.
Working cycle and airflow

Fig. 2-1  A comparison between the working cycle of a turbo-jet engine and a piston engine.

4. The working cycle upon which the gas turbine engine functions is, in its simplest form, represented by the cycle shown on the pressure volume diagram in Fig. 2-2. Point A represents air at atmospheric pressure that is compressed along the line AB. From B to C heat is added to the air by introducing and burning fuel at constant pressure, thereby considerably increasing the volume of air. Pressure losses in the combustion chambers (Part 4) are indicated by the drop between B and C. From C to D the gases resulting from combustion expand through the turbine and jet pipe back to atmosphere. During this part of the cycle, some of the energy in the expanding gases is turned into mechanical power.
Working cycle and airflow

![Diagram of engine cycle and airflow](image)

**Fig. 2-5-1** Airflow systems.
Fig. 3-7  Typical axial flow compressors.
efficiency, hence lower fuel consumption. For this reason the pure jet engine where all the airflow passes through the full compression cycle is now obsolete for all but the highest speed aircraft.

18. With the high by-pass ratio turbo-fan this trend is taken a stage further. The intake air undergoes only one stage of compression in the fan before being split between the core or gas generator system and the by-pass duct in the ratio of approximately one to five (fig. 3-8). This results in the optimum arrangement for passenger and/or transport aircraft flying at just below the speed of sound. The fan may be coupled to the front of a number of core compression stages (two shaft engine) or a separate shaft driven by its own turbine (three shaft engine).

Principles of operation

19. During operation the rotor is turned at high speed by the turbine so that air is continuously induced into the compressor, which is then accelerated by the rotating blades and swept rearwards onto the adjacent row of stator vanes. The pressure rise results from the energy imparted to the air in the rotor which increases the air velocity. The air is then decelerated (diffused) in the following stator passage and the kinetic energy translated into pressure. Stator vanes also serve to correct the deflection given to the air by the rotor blades and to present the air at the correct angle to the next stage of rotor blades. The last row of stator vanes usually act as air straighteners to remove swirl from the air prior to entry into the combustion system at a reasonably uniform axial velocity. Changes in pressure and velocity that occur in the airflow through the compressor are shown diagrammatically in fig. 3-9. The changes are accompanied by a progressive increase in air temperature as the pressure increases.

20. Across each stage the ratio of total pressures of outgoing air and inlet air is quite small, being between 1:1 and 1:2. The reason for the small pressure increase through each stage is that the rate of diffusion and the deflection angle of the blades must be limited if losses due to air breakaway at the blades and subsequent blade stall are to be avoided. Although the pressure ratio of each stage is small, every stage increases the exit pressure of the stage that precedes it. So whilst this first stage of a compressor may only increase the pressure by 3 to 4 lb. per sq.in., at the rear of a thirty to one compression system the stage pressure rise can be up to 80 lb. per sq.in. The ability to design multi-stage axial compressors with controlled air velocities and straight through flow minimizes losses and results in a high efficiency and hence low fuel consumption. This gives it a further advantage over the centrifugal compressor where these conditions are fundamentally not so easily achieved.

21. The more the pressure ratio of a compressor is increased the more difficult it becomes to ensure that it will operate efficiently over the full speed range. This is because the requirement for the ratio of inlet area to exit area, at the high speed case, results in an inlet area that becomes progressively too large relative to the exit area as the compressor speed and hence pressure ratio is reduced. The axial velocity of the inlet air in the front stages thus becomes low relative to the blade speed, this changes the incidence of the air onto the blades and a condition is reached where the flow separates and the compressor flow breaks down. Where high pressure ratios are required from a single compressor this problem can be overcome by introducing variable stator vanes in the front stages of the system. This corrects the incidence of air onto the rotor blades to angles which they can tolerate. An alternative is the incorporation of interstage bleed, where a proportion of air after entering the compressor is removed at an intermediate stage and dumped into the by-pass flow. While this method corrects the axial
velocity through the preceding stages energy is wasted and incorporation of variable stators is preferred.

22. The fan of the high by-pass ratio turbo-fan is an example of an axial compressor which has been optimized to meet the specific requirements of this cycle. While similar in principle to the core compressor stage, the proportions of design are such that the inner gas path is similar to that of the core compressor that follows it, while the tip diameter is considerably larger. The mass flow passed by the fan is typically six times that required by the core, the remaining five sixths by-pass the core and is expanded through its own coaxial nozzle, or may be mixed with the flow at exit from the core in a common nozzle. To optimize the cycle the by-pass flow has to be raised to a pressure of approximately 1.6 times the inlet pressure. This is achieved in the fan by utilizing very high tip speeds (1500 ft. per sec.) and airflow such that the by-pass section of the blades operate with a supersonic inlet air velocity of up to Mach 1.5 at the tip. The pressure that results is graded from a high value at the tip where relative velocities are highest to the more normal values of 1.3 to 1.4 at the inner radius which supercharges the core where aerodynamic design is more akin to that of a conventional compressor stage. The capability of this type of compressor stage achieves the cycle requirement of high flow per unit of frontal area, high efficiency and high pressure ratio in a single rotating blade row without inlet guide vanes within an acceptable engine diameter. Thus keeping weight and mechanical complexity at an acceptable level.

Construction
23. The construction of the compressor centres around the rotor assembly and casings. The rotor shaft is supported in ball and roller bearings and coupled to the turbine shaft in a manner that allows for any slight variation of alignment. The cylindrical casing assembly may consist of a number of cylindrical casings with a bolted axial joint between each stage or the casing may be in two halves with a bolted centre line joint. One or other of these construction methods is required in order that the casing can be assembled around the rotor.

Rotors
24. In compressor designs (fig. 3-10) the rotational speed is such that a disc is required to support the centrifugal blade load. Where a number of discs are fitted onto one shaft they may be coupled and secured together by a mechanical fixing but generally the discs are assembled and welded together, close to their periphery, thus forming an integral drum.

25. Typical methods of securing rotor blades to the disc are shown in fig. 3-11, fixing may be circumferential or axial to suit special requirements of the stage. In general the aim is to design a securing feature that imparts the lightest possible load on the supporting disc thus minimizing disc weight. Whilst most compressor designs have separate blades for manufacturing and maintainability requirements, it becomes more difficult on the smallest engines to design a practical fixing. However this may be overcome by producing blades integral with the disc; the so called 'blisk'.
Fig. 3-11  Methods of securing blades to disc.

**Rotor blades**

26. The rotor blades are of airfoil section (fig. 3-12) and usually designed to give a pressure gradient along their length to ensure that the air maintains a reasonably uniform axial velocity. The higher pressure towards the tip balances out the centrifugal action of the rotor on the airstream. To obtain these conditions, it is necessary to 'twist' the blade from root to tip to give the correct angle of incidence at each point. Air flowing through a compressor creates two boundary layers of slow to stagnant air on the inner and outer walls. In order to compensate for the slow air in the boundary layer a localized increase in blade camber both at the blade tip and root has been introduced. The blade extremities appear as if formed by bending over each corner, hence the term 'end-bend'.

**Stator vanes**

27. The stator vanes are again of airfoil section and are secured into the compressor casing or into stator vane retaining rings, which are themselves secured to the casing (fig. 3-13). The vanes are often assembled in segments in the front stages and may be shrouded at their inner ends to minimize the vibrational effect of flow variations on the longer vanes. It is also necessary to lock the stator vanes in such a manner that they will not rotate around the casing.
OPERATING CONDITIONS

28. Each stage of a multi-stage compressor possesses certain airflow characteristics that are dissimilar from those of its neighbour; thus to design a workable and efficient compressor, the characteristics of each stage must be carefully matched. This is a relatively simple process to implement for one set of conditions (design mass flow, pressure ratio and rotational speed), but is much more difficult when reasonable matching is to be retained with the compressor operating over a wide range of conditions such as an aircraft engine encounters.

29. If the operating conditions imposed upon the compressor blade depart too far from the design intention, breakdown of airflow and/or aerodynamically induced vibration will occur. These phenomena may take one of two forms; the blades may stall because the angle of incidence of the air relative to the blade is too high (positive incidence stall) or too low (negative incidence stall). The former is a front stage problem at low speeds and the latter usually affects the rear stages at high speed, either can lead to blade vibration which can induce rapid destruction. If the engine demands a pressure rise from the compressor which is higher than the blading can sustain, 'surge' occurs. In this case there is an instantaneous breakdown of flow through the machine and the high pressure air in the combustion system is expelled forward through the compressor with a loud 'bang' and a resultant loss of engine thrust.

Fig. 3-12 A typical rotor blade showing twisted contour.

Fig. 3-13 Methods of securing vanes to compressor casing.
COMBUSTION PROCESS

4. Air from the engine compressor enters the combustion chamber at a velocity up to 500 feet per second, but because at this velocity the air speed is far too high for combustion, the first thing that the chamber must do is to diffuse it, i.e. decelerate it and raise its static pressure. Since the speed of burning kerosine at normal mixture ratios is only a few feet per second, any fuel lit even in the diffused air stream, which now has a velocity of about 80 feet per second, would be blown away. A region of low axial velocity has therefore to be created in the chamber, so that the flame will remain alight throughout the range of engine operating conditions.

5. In normal operation, the overall air/fuel ratio of a combustion chamber can vary between 45:1 and 130:1. However, kerosine will only burn efficiently at, or close to, a ratio of 15:1, so the fuel must be burned with only part of the air entering the chamber, in what is called a primary combustion zone. This is achieved by means of a flame tube (combustion liner) that has various devices for metering the airflow distribution along the chamber.

6. Approximately 20 per cent of the air mass flow is taken in by the snout or entry section (fig. 4-2). Immediately downstream of the snout are swirl vanes and a perforated flare, through which air passes into the primary combustion zone. The swirling air induces a flow upstream of the centre of the flame tube and promotes the desired recirculation. The air not picked up by the snout flows into the annular space between the flame tube and the air casing.

7. Through the wall of the flame tube body, adjacent to the combustion zone, are a selected number of secondary holes through which a further 20 per cent of the main flow of air passes into the primary zone. The air from the swirl vanes and that from the secondary air holes interacts and creates a region of low velocity recirculation. This takes the form of a toroidal vortex, similar to a smoke ring, which has the effect of stabilizing and anchoring the flame (fig. 4-3). The recirculating gases hasten the burning of freshly
Fig. 4-2  Apportioning the airflow.

Injected fuel droplets by rapidly bringing them to ignition temperature.

8. It is arranged that the conical fuel spray from the nozzle intersects the recirculation vortex at its centre. This action, together with the general turbulence in the primary zone, greatly assists in breaking up the fuel and mixing it with the incoming air.

9. The temperature of the gases released by combustion is about 1,800 to 2,000 deg.C., which is far too hot for entry to the nozzle guide vanes of the turbine. The air not used for combustion, which amounts to about 60 per cent of the total airflow, is therefore introduced progressively into the flame tube. Approximately a third of this is used to lower the gas temperature in the dilution zone before it enters the turbine and the remainder is used for cooling the walls of the flame tube. This is achieved by a film of cooling air flowing along the inside surface of the flame tube wall, insulating it from the hot combustion gases (fig. 4-4). A recent development allows cooling air to enter a network of passages within the flame tube wall before exiting to form an insulating film of air, this can reduce the required wall cooling airflow by up to 50 per cent. Combustion should be completed before the dilution air enters the flame tube, otherwise the incoming air will cool the flame and incomplete combustion will result.

10. An electric spark from an igniter plug (Part 11) initiates combustion and the flame is then self-sustained.

Fig. 4-3  Flame stabilizing and general airflow pattern.
Fig. 4-8 Tubo-annular combustion chamber.

Combustion by aerating the over-rich pockets of fuel vapours close to the spray nozzle; this results in a large reduction in initial carbon formation.

COMBUSTION CHAMBER PERFORMANCE

23. A combustion chamber must be capable of allowing fuel to burn efficiently over a wide range of operating conditions without incurring a large pressure loss. In addition, if flame extinction occurs, then it must be possible to relight. In performing these functions, the flame tube and spray nozzle atomizer components must be mechanically reliable.

24. The gas turbine engine operates on a constant pressure cycle, therefore any loss of pressure during the process of combustion must be kept to a minimum. In providing adequate turbulence and mixing, a total pressure loss varying from about 3 to 8 per cent of the air pressure at entry to the chamber is incurred.

Combustion intensity

25. The heat released by a combustion chamber or any other heat generating unit is dependent on the volume of the combustion area. Thus, to obtain the required high power output, a comparatively small and
Fig. 5-1  A triple-stage turbine with single shaft system.

gas flow, the rotational speed at which it must be produced and the diameter of turbine permitted.

3. The number of shafts, and therefore turbines, varies with the type of engine; high compression ratio engines usually have two shafts, driving high and low pressure compressors (fig. 5-2). On high by-pass ratio fan engines that feature an intermediate pressure system, another turbine may be interposed between the high and low pressure turbines, thus forming a triple-spool system (fig. 5-3). On some engines, driving torque is derived from a free-power turbine (fig. 5-4). This method allows the turbine to run at its optimum speed because it is mechanically independent of other turbine and compressor shafts.
Fig. 5-5  Comparison between a pure impulse turbine and an impulse/reaction turbine.

and the turbine. This transfer is never 100 per cent because of thermodynamic and mechanical losses, (para. 11).

7. When the gas is expanded by the combustion process (Part 4), it forces its way into the discharge nozzles of the turbine where, because of their convergent shape, it is accelerated to about the speed of sound which, at the gas temperature, is about 2,500 feet per second. At the same time the gas flow is given a 'spin' or 'whirl' in the direction of rotation of the turbine blades by the nozzle guide vanes. On impact with the blades and during the subsequent reaction through the blades, energy is absorbed, causing the turbine to rotate at high speed and so provide the power for driving the turbine shaft and compressor.

8. The torque or turning power applied to the turbine is governed by the rate of gas flow and the energy change of the gas between the inlet and the outlet of the turbine blades. The design of the turbine is such that the whirl will be removed from the gas stream so that the flow at exit from the turbine will be substantially 'straightened out' to give an axial flow into the exhaust system (Part 6). Excessive residual whirl reduces the efficiency of the exhaust system and also tends to produce jet pipe vibration which has a detrimental effect on the exhaust cone supports and struts.

9. It will be seen that the nozzle guide vanes and blades of the turbine are 'twisted', the blades having a stagger angle that is greater at the tip than at the root (fig. 5-6). The reason for the twist is to make the gas flow from the combustion system do equal work at all positions along the length of the blade and to ensure that the flow enters the exhaust system with a uniform axial velocity. This results in certain changes in velocity, pressure and temperature occurring through the turbine, as shown diagrammatically in fig. 5-7.

10. The 'degree of reaction' varies from root to tip, being least at the root and highest at the tip, with the mean section having the chosen value of about 50 per cent.
11. The losses which prevent the turbine from being 100 per cent efficient are due to a number of reasons. A typical uncooled three-stage turbine would suffer a 3.5 per cent loss because of aerodynamic losses in the turbine blades. A further 4.5 per cent loss would be incurred by aerodynamic losses in the nozzle guide vanes, gas leakage over the turbine blade tips and exhaust system losses; these losses are of approximately equal proportions. The total losses result in an overall efficiency of approximately 92 per cent.

CONSTRUCTION

12. The basic components of the turbine are the combustion discharge nozzles, the nozzle guide vanes, the turbine discs and the turbine blades. The rotating assembly is carried on bearings mounted in the turbine casing and the turbine shaft may be common to the compressor shaft or connected to it by a self-aligning coupling.

Nozzle guide vanes

13. The nozzle guide vanes are of an aerofoil shape with the passage between adjacent vanes forming a convergent duct. The vanes are located (fig. 5-8) in the turbine casing in a manner that allows for expansion.
Fig. 5-12 Various turbine blade crystal structures.
27. In the past, turbine discs have been made in ferritic and austenitic steels but nickel based alloys are currently used. Increasing the alloying elements in nickel extend the life limits of a disc by increasing fatigue resistance. Alternatively, expensive powder metallurgy discs, which offer an additional 10% in strength, allow faster rotational speeds to be achieved.

**Turbine blades**

28. A brief mention of some of the points to be considered in connection with turbine blade design will give an idea of the importance of the correct choice of blade material. The blades, while glowing red-hot, must be strong enough to carry the centrifugal loads due to rotation at high speed. A small turbine blade weighing only two ounces may exert a load of over two tons at top speed and it must withstand the high bending loads applied by the gas to produce the many thousands of turbine horse-power necessary to drive the compressor. Turbine blades must also be resistant to fatigue and thermal shock, so that they will not fail under the influence of high frequency fluctuations in the gas conditions, and they must also be resistant to corrosion and oxidation. In spite of all these demands, the blades must be made in a material that can be accurately formed and machined by current manufacturing methods.

29. From the foregoing, it follows that for a particular blade material and an acceptable safe life there is an associated maximum permissible turbine entry temperature and a corresponding maximum engine power. It is not surprising, therefore, that metallurgists and designers are constantly searching for better turbine blade materials and improved methods of blade cooling.

30. Over a period of operational time the turbine blades slowly grow in length. This phenomenon is known as 'creep' and there is a finite useful life limit before failure occurs.

31. The early materials used were high temperature steel forgings, but these were rapidly replaced by cast nickel base alloys which give better creep and fatigue properties.

32. Close examination of a conventional turbine blade reveals a myriad of crystals that lie in all directions (equi-axed). Improved service life can be obtained by aligning the crystals to form columns along the blade length, produced by a method known as 'Directional Solidification'. A further advance of this technique is to make the blade out of a single crystal. Examples of these structures are shown in fig. 5-12. Each
Fig. 7-1  Mechanical arrangement of accessory drives.
Pressure relief valve system

5. In the pressure relief valve system the oil flow to the bearing chambers is controlled by limiting the pressure in the feed line to a given design value. This is accomplished by the use of a spring loaded valve which allows oil to be directly returned from the pressure pump outlet to the oil tank, or pressure pump inlet, when the design value is exceeded. The valve opens at a pressure which corresponds to the idling speed of the engine, thus giving a constant feed pressure over normal engine operating speeds. However, increasing engine speed causes the bearing chamber pressure to rise sharply. This reduces the pressure difference between the bearing chamber and feed jet, thus decreasing the oil flow rate to the bearings as engine speed increases. To alleviate this problem, some pressure relief valve systems use the increasing bearing chamber pressure to augment the relief valve spring load. This maintains a constant flow rate at the higher engine speeds by increasing the pressure in the feed line as the bearing chamber pressure increases.

6. Fig. 8-1 shows the pressure relief valve system for a turbo-propeller engine and indicates the basic components that comprise an engine lubrication system. The oil pressure pump draws oil from the tank through a strainer which protects the pump gears from debris which may have entered the tank. Oil is then delivered through a pressure filter to the pressure relief valve which maintains a constant oil delivery pressure to the feed jets in the bearing chambers. Some engines may have an additional relief valve (pressure limiting valve) which is fitted at the oil pressure pump outlet. This valve is set to open at a much higher value than the pressure relief valve to return the oil to the inlet side of the oil pressure pump in the event of the system becoming blocked. A similar valve may also be fitted across the pressure filter to prevent oil starvation of the bearing chambers should the filter become partially blocked or the oil having a high viscosity under cold starting conditions preventing sufficient flow through the filter. Provision is also made to supply oil to the propeller pitch control system, reduction gear and

Fig. 8-1 A pressure relief valve type oil system.
torquemeter system. Scavenge pumps return the oil to the tank via the oil cooler. On entering the tank, the oil is de-aerated ready for recirculation.

**Full flow system**

7. Although the pressure relief valve system operates satisfactorily for engines which have a low bearing chamber pressure, which does not unduly increase with engine speed, it becomes an undesirable system for engines which have high chamber pressures. For example, if a bearing chamber has a maximum pressure of 90 lb per sq. in. It would require a pressure relief valve setting of 130 lb per sq. in. to produce a pressure drop of 40 lb per sq. in. at the oil feed jet. This results in the need for large pumps and difficulty in matching the required oil flow at slower speeds.

8. The full flow system achieves the desired oil flow rates throughout the complete engine speed range by dispensing with the pressure relief valve and allowing the pressure pump delivery pressure to supply directly the oil feed jets. Fig. 8-2 shows an example of this system which may be found on a turbo-fan engine. The pressure pump size is determined by the flow required at maximum engine speed. The use of this system allows smaller pressure and scavenge pumps to be used since the large volume of oil which is spilled by the pressure relief valve system at maximum engine speed is obviated.

![Diagram of an oil system](image)

**Fig. 8-2** A full flow type oil system.
pressure and temperature. Therefore, to reduce engine performance losses, the air is taken as early as possible from the compressor commensurate with the requirement of each particular function. The cooling air is expelled overboard via a vent system or into the engine main gas stream, at the highest possible pressure, where a small performance recovery is achieved.

COOLING

3. An important consideration at the design stage of a gas turbine engine is the need to ensure that certain parts of the engine, and in some instances certain accessories, do not absorb heat to the extent that is detrimental to their safe operation. The principal areas which require air cooling are the combustor and turbine. Refer to Part 4 for combustor cooling techniques.

4. Cooling air is used to control the temperature of the compressor shafts and discs by either cooling or heating them. This ensures an even temperature distribution and therefore improves engine efficiency by controlling thermal growth and thus maintaining minimum blade tip and seal clearances. Typical cooling and sealing airflows are shown in fig. 9-1.

Turbine cooling

5. High thermal efficiency is dependent upon high turbine entry temperature, which is limited by the turbine blade and nozzle guide vane materials. Continuous cooling of these components allows their environmental operating temperature to exceed the material's melting point without affecting the blade and vane integrity. Heat conduction from the turbine blades to the turbine disc requires the discs to be cooled and thus prevent thermal fatigue and uncontrolled expansion and contraction rates.

6. An air cooled high pressure nozzle guide vane and turbine blade arrangement illustrating the cooling

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Fig. 9-1  General internal airflow pattern.
Fuel systems

The dotted line represents the sensing signal from the engine.

Fig. 10-3  Simplified fuel systems for turbo-propeller and turbo-jet engines.
4. On propeller-powered aircraft, reverse thrust action is obtained by changing the pitch of the propeller blades. This is usually achieved by a hydro-mechanical system, which changes the blade angle to give the braking action under the response of the power or throttle lever in the aircraft.

5. Ideally, the gas should be directed in a completely forward direction. It is not possible, however, to achieve this, mainly for aerodynamic reasons, and a discharge angle of approximately 45 degrees is chosen. Therefore, the effective power in reverse thrust is proportionately less than the power in forward thrust for the same throttle angle.

**PRINCIPLES OF OPERATION**

6. There are several methods of obtaining reverse thrust on turbo-jet engines; three of these are shown in fig. 15-2 and explained in the following paragraphs.

7. One method uses clamshell-type deflector doors to reverse the exhaust gas stream and a second uses a target system with external type doors to do the same thing. The third method used on fan engines utilizes blocker doors to reverse the cold stream airflow.

8. Methods of reverse thrust selection and the safety features incorporated in each system described are basically the same. A reverse thrust lever in the crew compartment is used to select reverse thrust; the lever cannot be moved to the reverse thrust position unless the engine is running at a low power setting, and the engine cannot be opened up to a high power setting if the reverser fails to move into the full reverse thrust position. Should the operating pressure fail or fall, a mechanical lock holds the reverser in the forward thrust position; this lock cannot be removed until the pressure is restored. Operation of the thrust reverser system is indicated in the crew compartment by a series of lights.

**Clamshell door system**

9. The clamshell door system is a pneumatically operated system, as shown in detail in fig. 15-3. Normal engine operation is not affected by the system, because the ducts through which the exhaust gases are deflected remain closed by the doors until reverse thrust is selected by the pilot.

10. On the selection of reverse thrust, the doors rotate to uncover the ducts and close the normal gas stream exit. Cascade vanes then direct the gas stream in a forward direction so that the jet thrust opposes the aircraft motion.
Thrust reversal

Fig. 15-2   Methods of thrust reversal.
Thrust reversal

Fig. 15-6  Hot stream thrust reverser installations.
Fig. 16-1  Principle of afterburning

4. The area of the afterburning jet pipe is larger than a normal jet pipe would be for the same engine, to obtain a reduced velocity gas stream. To provide for operation under all conditions, an afterburning jet pipe is fitted with either a two-position or a variable-area propelling nozzle (fig. 16-2). The nozzle is closed during non-afterburning operation, but when afterburning is selected the gas temperature increases and the nozzle opens to give an exit area suitable for the resultant increase in the volume of the gas stream. This prevents any increase in pressure occurring in the jet pipe which would affect the functioning of the engine and enables afterburning to be used over a wide range of engine speeds.

5. The thrust of an afterburning engine, without afterburning in operation, is slightly less than that of a similar engine not fitted with afterburning equipment; this is due to the added restrictions in the jet pipe. The overall weight of the power plant is also increased because of the heavier jet pipe and afterburning equipment.

6. Afterburning is achieved on low by-pass engines by mixing the by-pass and turbine streams before the afterburner fuel injection and stabilizer system is reached so that the combustion takes place in the mixed exhaust stream. An alternative method is to inject the fuel and stabilize the flame in the individual by-pass and turbine streams, burning the available gases up to a common exit temperature at the final nozzle. In this method, the fuel injection is scheduled separately to the individual streams and it is normal to provide some form of interconnection between the flame stabilizers in the hot and cold streams to assist the combustion processes in the cold by-pass air.

OPERATION OF AFTERBURNING

7. The gas stream from the engine turbine enters the jet pipe at a velocity of 750 to 1,200 feet per second, but as this velocity is far too high for a stable flame to be maintained, the flow is diffused before it enters the afterburner combustion zone, i.e. the flow velocity is reduced and the pressure is increased. However, as the speed of burning kerosine at normal mixture ratios is only a few feet per second, any fuel lit even in the diffused air stream would be blown away. A form of flame stabilizer (vapour gutter) is, therefore, located downstream of the fuel burners to provide a region in which turbulent eddies are formed to assist combustion and where the local gas velocity is further reduced to a figure at which flame stabilization occurs whilst combustion is in operation.
22. It is not possible to go on increasing the amount of fuel that is burnt in the jet pipe so that all the available oxygen is used, because the jet pipe would not withstand the high temperatures that would be incurred and complete combustion cannot be assured.

**FUEL CONSUMPTION**

23. Afterburning always incurs an increase in specific fuel consumption and is, therefore, generally limited to periods of short duration. Additional fuel must be added to the gas stream to obtain the required temperature ratio (para. 19). Since the temperature rise does not occur at the peak of compression, the fuel is not burnt as efficiently as in the engine combustion chamber and a higher specific fuel consumption must result. For example, assuming a specific fuel consumption without afterburning of 1.15 lb/hr/lb thrust at sea level and a speed of Mach 0.9 as shown in fig. 16-9, then with 70 per cent afterburning under the same conditions of flight, the consumption will be increased to

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**Fig. 16-9** Specific fuel consumption comparison.

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**Fig. 16-10** Afterburning and its effect on the rate of climb.
Thrust distribution

Forward gas load: 57,836 lb.
Rearward gas load: 46,678 lb.
Total thrust: 11,158 lb.

19,049 lb.
2,186 lb.
34,182 lb.
41,091 lb.
5,587 lb.
2,419 lb.

Fig. 20-1 Thrust distribution of a typical single-spool axial flow engine.

3. At the start of the cycle, air is induced into the engine and is compressed. The rearward accelerations through the compressor stages and the resultant pressure rise produces a large reactive force in a forward direction. On the next stage of its journey the air passes through the diffuser where it exerts a small reactive force, also in a forward direction.

4. From the diffuser the air passes into the combustion chambers (Part 4) where it is heated, and in the consequent expansion and acceleration of the gas large forward forces are exerted on the chamber walls.

5. When the expanding gases leave the combustion chambers and flow through the nozzle guide vanes they are accelerated and deflected on to the blades of the turbine (Part 5). Due to the acceleration and deflection, together with the subsequent straightening of the gas flow as it enters the jet pipe, considerable 'drag' results; thus the vanes and blades are subjected to large rearward forces, the magnitude of which may be seen on the diagram. As the gas flow passes through the exhaust system (Part 6), small forward forces may act on the inner cone or bullet, but generally only rearward forces are produced and these are due to the 'drag' of the gas flow at the propelling nozzle.

6. It will be seen that during the passage of the air through the engine, changes in its velocity and pressure occur (Part 2). For instance, where a conversion from velocity (kinetic) energy to pressure energy is required the passages are divergent in shape, similar to that used in the compressor diffuser. Conversely, where it is required to convert the energy stored in the combustion gases to velocity, a convergent passage or nozzle, similar to that used in the turbine, is employed. Where the conversion is to velocity energy, 'drag' loads or rearward forces are produced; where the conversion is to pressure energy, forward forces are produced. Part 2, fig. 2-3 illustrates velocity and pressure changes at two points on the engine.